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# Applications of Endothermic Reaction Technology to HSCT and AST

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## SUMMARY

The success of strategies for controlling emissions and enhancing performance in High Speed Research and Advanced Subsonic Technology applications may be increased by more effective utilization of the heat sink afforded by the fuel in the vehicle thermal management system. This study quantifies the potential benefits associated with the use of supercritical preheating and endothermic cracking of jet fuel prior to combustion to enhance the thermal management capabilities of the propulsion systems in the High Speed Civil Transport (HSCT) and the Advanced Subsonic Transport (AST). A fuel-cooled thermal management system, consisting of plate-fin heat exchangers and a small auxiliary compressor, is defined for the HSCT, integrated with the engine, and an assessment of the effect on engine performance, weight, and operating cost is performed. The analysis indicates significant savings due a projected improvement in fuel economy, and the potential for additional benefit if the cycle is modified to take full advantage of all the heat sink available in the fuel. Critical requirements for system development are identified, and an approach for validating the concept is described. An abridged version of this report, approved for public release, is available as Reference 1.

## INTRODUCTION

The HSCT engine will operate at near maximum temperature and stress levels throughout supersonic cruise. Therefore, although the cycle temperatures are not much different from current generation engines, the time at temperature (hot time) and stress will be up to 30 times greater. In contrast, the AST engine will operate at much higher overall pressure ratio than current engines, with the maximum compressor exit temperature and stress produced at takeoff. In either of these applications, very significant technical challenges must be resolved to advance the development of a reliable and durable engine.

Advanced materials development programs are giving rise to new nickel alloys for the compressor and turbine disks, new nickel and thermal barrier coating (TBC) systems for the turbine airfoils, and ceramic matrix composites (CMC) for the combustor liner and exhaust nozzle. Current efforts have provided confidence in the ability to meet the minimum acceptable life goals for those parts which have been analyzed in detail. However, an increase in reliability and durability over the minimum acceptable would have a significant payoff.

A fuel-cooled thermal management system would prove very valuable when used to reduce the temperature of the turbine airfoil and disk cooling air. Compressor bore cooling is also an attractive application. In either case, disk radial temperature profiles (i.e., thermal stresses) need to be considered and the level of cooling selected upon consideration of a possible impact on weight. Similarly, an endothermic reaction system might prove beneficial for cooling the bearing compartments or electronic control systems, and waste heat absorbed by the fuel can be returned to the cycle, improving performance. In addition, injection and combustion of supercritical vaporized fuel would eliminate the current requirement for recirculating fuel back to the tank. Moreover, supercritical preheating and endothermic cracking of the fuel prior to combustion may enable improved mixing, broadened combustion stability, and reduced smoke and $NO_X$ emissions.

## HSCT CYCLE ANALYSIS

The baseline engine cycle for powering a Mach 2.4 commercial transport has been selected for this study. The engine is an 800 lb/sec two-spool turbofan which incorporates a three-stage (3.7 pressure ratio) fan, a five-stage (5.2 pressure ratio) high pressure compressor, a single-stage high pressure turbine, and a two-stage low pressure turbine.

Initial efforts in this study were directed at determining the impact on engine performance of cooling the turbine airfoil and disk with preconditioned compressor bleed air. Analyses of the engine cycle and fuel thermal management system were conducted at the Mach number 2.4 cruise condition, which represents the most severe thermal environment. Table 1 summarizes the engine operation at three different flight conditions.

**Table 1 - Summary of HSCT Engine Conditions for Thermal Analysis**

| Condition | Begin Cruise | Begin Descent | Minimum Fuel Flow |
|---|---|---|---|
| Mach Number | 2.4 | 2.4 | 1.3 |
| Altitude, ft | 55,000 | 65,000 | 50,000 |
| Engine Power Setting | Max. Power | Idle | Idle |
| Compressor Exit Temperature, F | 1200 | 1040 | 510 |
| Fuel Flow, lbm/hr | 29140 | 9270 | 2900 |
| Combustor Exit Temperature, F | 2970 | 2305 | 1400 |
| High Turbine Rotor Inlet Temperature, F | 2800 | 2180 | 1310 |
| Net Thrust, lbf | 23870 | 7360 | 2400 |
| TSFC, lbm/hr-lbf | 1.22 | 1.26 | 1.21 |

A conceptual approach for using the endothermic heat sink potential of the fuel was defined, and a preliminary architecture was configured for a corresponding thermal management system. The system is configured to provide means for cooling a portion of the high-pressure compressor discharge air, using part of the engine fuel flow as the coolant. The cooled air is subsequently used to cool the high-pressure-turbine airfoils, with a portion being directed through hollow vanes across the compressor diffuser, into the engine bore.

Evaluation of the potential benefits achievable requires the ability to design and evaluate both the engine cycle and the thermal management system. Pratt & Whitney (P&W) currently uses a proprietary *State of the Art Performance Program* (SOAPP) to model and evaluate gas turbine engine cycle performance for the HSCT. Thermal management systems are modeled and evaluated using another P&W code, known as the *Fuel and Integrated Thermal System Simulator* (FITSS) code. At the present time the FITSS model depends on input from the SOAPP performance code to define the engine interface conditions. Therefore, the two modeling systems are used interactively in an iterative procedure to evaluate thermal system and engine cycle performance.

A conceptual approach for the thermal management system was selected, in which fuel would be used to cool compressor bleed air, with the cooled air subsequently being used to cool the turbine. The specific strategy for using this cooling resource was selected from among the following possibilities:

- Reducing the amount of cooling flow, with constant metal temperature and burner states,
- Reducing the metal temperature, with constant flow and burner states, and
- Increasing the burner temperature/pressure, with constant cooling flow and metal temperature.

These various combinations can result in reduced thrust specific fuel consumption (TSFC), increased engine hot section life, increased engine thrust-to-weight ratio, or combinations of the foregoing. Since increasing burner temperature implies an accompanying increase in $NO_x$ emissions, this option was eliminated from further consideration. Several of these trades have been examined, and the corresponding thermal management system designed. The required analysis for consideration of performance trades involves only the engine cycle, as long as the assumed performance of the thermal management system is achievable.

The initial evaluation of the effect of the thermal management concept was performed at the beginning of cruise (55,000-ft, Mach number 2.4), with the engine at maximum power. The extracted bleed air (16.5% of the core flow) was assumed to be cooled with 80% of the engine fuel flow, yielding an air temperature reduction of approximately 200 F in the turbine cooling air. This level of cooling was specified to avoid generating large thermal gradients that would increase the stress and require a corresponding increase in component weight. Preliminary analysis indicated that this is an achievable cooling goal.

Table 2 shows the effect of fuel cooling on HSCT engine performance for two possible approaches, corresponding respectively to increased turbine lifetime and improved fuel economy. In both cases, the cooling air temperature is lowered 200 F, the high-pressure-turbine (HPT) rotor inlet temperature is maintained constant at 2800 F and the fuel temperature and heating value are increased (due to cracking reactions), permitting a reduction in the fuel flow rate. For increased lifetime, the amount of HPT cooling is maintained at the baseline flow rate of 16.5%, resulting in lower metal temperatures. For improved fuel economy, the HPT cooling is reduced to 12.4% while metal temperatures and, therefore, turbine life remain unchanged from the baseline engine.

**Table 2 - Effect of Fuel Cooling on HSCT Engine Performance at M = 2.4 (Cruise, 55000 ft)**

| | Baseline | Increased Lifetime | Improved Fuel Economy |
|---|---|---|---|
| Compressor Exit Temperature ($T_3$), F | 1200 | 1200 | 1200 |
| HPT Cooling Air, % | 16.5 | 16.5 | 12.4 |
| HPT Cooling Air Temperature, F | 1200 | 1000 | 1000 |
| Fuel Heating Value, Btu/lb | 18,580 | 18,950 | 18,850 |
| Burner Inlet Air Flow, Lb/sec | 260 | 261 | 263 |
| Fuel Flow, Lbm/hr | 29,110 | 28,980 | 28,325 |
| Fuel/Air Ratio | 0.0311 | 0.0308 | 0.0299 |
| Combustor Exit Temperature ($T_4$), F | 2970 | 2990 | 2935 |
| HPT Rotor Inlet Temperature ($T_{4.1}$), F | 2800 | 2800 | 2800 |
| LPT Inlet Temperature, F | 2200 | 2175 | 2217 |
| HPT Pressure Ratio | 2.46 | 2.48 | 2.39 |
| LPT Pressure Ratio | 1.90 | 1.90 | 1.95 |
| Nozzle Inlet Temperature, F | 1392 | 1388 | 1367 |
| Nozzle Inlet Pressure, psia | 36.5 | 36.56 | 36.7 |
| Net Thrust, Lbf | 23,850 | 23,780 | 23,410 |
| TSFC, Lbm/hr-lbf | 1.221 | 1.219 | 1.210 |
| Fuel Heating Value Change, % | ---- | + 2.0 | + 1.4 |
| Fuel Flow Change, % | ---- | - 0.4 | - 2.7 |
| Net Thrust Change, % | ---- | - 0.3 | - 1.9 |
| TSFC Change, % | ---- | - 0.1 | - 0.7 |

In the case corresponding to increased lifetime, the enhanced cooling allows the combustor exit temperature ($T_4$) to be increased by 20 F. There is a small reduction in the net thrust (0.3%) and a slight reduction in TSFC (0.1%) due to the combined effects of reducing the fuel flow and diverting energy from the bleed air to the fuel. When the benefit of fuel cooling is applied toward reducing the amount of HPT cooling air, then the engine bypass ratio can be increased from 0.416 to 0.475 at the sea level static design point, resulting in a 0.7% TSFC benefit at the supersonic cruise condition. It should be noted that for either case, the heat transferred to the fuel represents less than 50% of its heat sink potential and, therefore, ancillary cooling concepts are also possible.

A further study was performed to evaluate the cooling concept impact at subsonic speeds. The impact of reducing the HPT vane and blade cooling air temperature by 200 F was evaluated at the Mach number 0.9, 36,000-ft subsonic cruise condition. Table 3 shows the effect of fuel cooling on HSCT engine performance for the two approaches. As previously, for increased lifetime the amount of high-pressure-turbine cooling is maintained at the baseline flow rate of 16.5%, resulting in lower metal temperatures. For improved fuel economy, the HPT cooling is reduced to 12.4% while metal temperatures and, therefore, turbine life remain unchanged from the baseline engine. In both instances, the HPT rotor inlet temperature is maintained constant at 2800 F.

**Table 3 - Effect of Fuel Cooling on HSCT Engine Performance at M = 0.9 (Cruise, 36,000 ft)**

| | Baseline | Increased Lifetime | Improved Fuel Economy |
|---|---|---|---|
| HPT Cooling Air, % | 16.5 | 16.5 | 12.4 |
| Fuel Heating Value, Btu/lb | 18,580 | 19,230 | 19,055 |
| Net Thrust, lbf | 9160 | 9160 | 9160 |
| Fuel Flow, lbm/hr | 8090 | 8050 | 7960 |
| TSFC, lbm/hr-lbf | 0.883 | 0.879 | 0.860 |
| TSFC Change, % | Base | - 0.4 | - 1.6 |

In the case directed toward increasing lifetime, the heat added to the fuel from cooling the turbine cooling air increased the lower heating value of the fuel to 19,230 Btu/lb and reduced the specific fuel consumption by 0.4%. In the case aimed at improving fuel economy, the reduction in HPT cooling air from 16.5 to 12.4% was used to increase the engine bypass ratio from 0.416 to 0.475 at the sea level static design point. This cycle produced a 1.6% reduction in subsonic cruise fuel consumption.

An additional analysis was performed to evaluate the effects of the catalytic heat exchanger/reactor (CHER) design on component weight, pressure drop and thermal performance. Both the heat exchanger size and internal core geometry were varied to allow a determination of the relationship between weight and pressure loss, at a constant thermal effectiveness. The results were used to identify a design concept that will limit the maximum air-side pressure loss to approximately 5%. The resulting weight of the heat exchanger core will be approximately 160 lbm, representing an approximate 1% increase in the engine weight. The impact of increased weight is considered in the economic analysis performed as part of this study.

## AST CYCLE ANALYSIS

This analysis was performed by NASA LeRC with guidance from UTC regarding the baseline engine to be analyzed and the recommended engine modifications. Pratt & Whitney performed the engine thermal management analysis using the same models that were employed for the HSCT.

The baseline AST engine cycle selected for analysis in this study is the PW STS1046 Advanced Ducted Propulsor engine, as defined for the NASA Large Engine Technology study. (This study determined the critical or enabling technologies required for a year 2005 entry-into-service engine for subsonic commercial aircraft satisfying NASA's Advanced Subsonic Transport goals.) The STS1046 has a six-stage, 9:1 pressure ratio high compressor driven by a single-stage high turbine, and a six-stage variable geometry low compressor driven by a six-stage low turbine. The engine has been supercharged to achieve an overall-pressure-ratio (OPR) of 55 at the maximum rate of climb. Combustor exit temperature at takeoff is 3100 F. The propulsor is a geared, variable pitch fan with a cruise fan pressure ratio of 1.32. A fan diameter of 119-in. was selected to produce a favorable duct/engine exhaust velocity ratio. Table 4 summarizes the performance of the engine at several flight conditions.

**Table 4 - STS 1046 ADP/AST Baseline Performance**

| Condition | Takeoff | Max. Climb | Cruise |
|---|---|---|---|
| Mach Number | 0.2 | 0.85 | 0.85 |
| Altitude, ft | sea level | 35,000 | 35,000 |
| OPR | 48.4 | 54.9 | 52.6 |
| Compressor Exit Temperature, F | 1280 | 1147 | 1059 |
| Fuel Flow, lbm/hr | 13216 | 5582 | 5066 |
| Combustor Exit Temperature, F | 3100 | 2983 | 2786 |
| Net Thrust, lbf | 43436 | 10536 | 9900 |
| TSFC, lbm/hr-lbf | 0.304 | 0.530 | 0.512 |

Using the fuel as a heat sink, to precool the turbine cooling air, presents a cycle resource that can be used in several different ways. Two possible approaches that were considered for the AST are:

1) Reduce the turbine coolant flow rate while holding metal temperatures constant, and increase the bypass ratio, thereby reducing TSFC for equivalent thrust.
2) Hold coolant flow rate constant, thereby lowering metal temperatures and improving life.

**Table 5 - Effect of Fuel Cooling on AST Engine Performance (Cruise at M = 0.85, 35000 ft)**

| | Baseline | Improved Fuel Economy |
|---|---|---|
| Compressor Exit Temperature ($T_3$), °F | 985 | 989 |
| HPT Cooling Air, % core flow | 20.64 | 15.6 |
| HPT Cooling Air Temperature, °F | 989 | 789 |
| Fuel Heating Value, Btu/lbm | 18580 | 18787 |
| Burner Inlet Air Flow, lbm/sec | 42.07 | 41.99 |
| Fuel Flow, Lbm/hr | 3992.2 | 3922.7 |
| Fuel/Air Ratio | 0.02636 | 0.02595 |
| Combustor Exit Temperature ($T_4$), °F | 2546 | 2559 |
| HPT Rotor Inlet Temperature ($T_{4.1}$), °F | 2377 | 2419 |
| LPT Inlet Temperature, °F | 1617 | 1668 |
| HPT Pressure Ratio | 3.777 | 3.499 |
| LPT Pressure Ratio | 12.60 | 13.65 |
| Nozzle Inlet Temperature, °F | 716 | 727 |
| Nozzle Inlet Pressure, psia | 4.95 | 4.95 |
| Net Thrust, lbf | 8072.3 | 8074.1 |
| TSFC, lbm/hr-lbf | 0.4946 | 0.4858 |
| Fuel Heating Value Change, % | ---- | +1.1 |
| Fuel Flow Change, % | ---- | -1.7 |
| Net Thrust Change, % | ---- | +.02 |
| TSFC Change, % | ---- | -1.8 |

The first approach, i.e., reducing turbine cooling air to improve fuel economy, was examined and the results of the cycle analysis are shown in Table 5. In this method, the cooling air for the high-pressure turbine is cooled 200 F by the fuel prior to use in the turbine. The net effect of the reduced temperature coolant is that the cooling flow rate can be reduced from 20 to 15 percent of the core flow, while maintaining turbine metal temperatures constant. Since the coolant injection temperatures will remain approximately constant, reducing the coolant flow results in an increase in the temperatures in the turbine gas stream, and a redistribution of loading between the high and low pressure turbines. The heat absorbed by the fuel increases the effective heating value (+1.1%), thereby reducing the fuel flow rate that is required to achieve the rated thrust level. The net effect on the cycle is a decrease of 1.8% in TSFC at the cruise condition.

Life is not a driver for the AST engine because of the much less severe operating condition. On the other hand, specific fuel consumption (SFC) is a strong economic driver. Therefore, the approach yielding reduced SFC was selected for preliminary evaluation of the AST.

A corresponding study was performed to evaluate the effects of the catalytic heat exchanger/reactor (CHER) design on component weight, pressure drop, and thermal performance. A CHER design was identified that will satisfy the air cooling requirement at the selected cruise condition. The weight of the CHER core would be approximately 218 lbm (not including manifolds and other external plumbing), and result in an air-side pressure loss of 1.8%. The total loss in turbine cooling air pressure across the entire thermal management system is estimated to be less than 5%.

## CONCEPT DEFINITION

At this point in the study it was planned that the two aircraft applications and relevant benefits be reviewed, and a selection by NASA of the single most-promising application be made on which to concentrate further detailed studies. Although the potential benefits associated with using fuel to precool the turbine cooling air were determined to be significant for both applications, the NASA review concluded that there seemed to be greater potential and higher payoff associated with the HSCT cycle application. Consequently, a decision was made to defer further AST analyses and proceed with more detailed conceptual design studies for the HSCT that would include a schematic layout of the integrated system and an estimate of the weight change relative to the baseline engine.

Based on the HSCT cycle analysis, the thermal management concept selected for increasing component life in the engine hot section involves cooling 16.5% of the high-pressure-compressor discharge air ($T_3$) with 80% of the fuel in an external catalytic heat exchanger reactor (CHER). The results of the initial evaluation indicate that the desired 200 F bleed air temperature reduction can be attained, and that all turbine components can be cooled with air at the same condition. The CHER pressure loss depends on the size of the heat exchanger, and the analysis showed that a CHER core weighing approximately 160 lb could provide the desired cooling with a total pressure loss of approximately 5-6% (including plumbing losses).

Current and state-of-the-art turbine cooling concepts employ different methods for cooling various stages in the turbine, with very significant differences between the first vane and subsequent downstream stages. Traditional vane cooling concepts utilize upstream-facing orifices in the leading edge (showerhead cooling concept) to inject a film of cooling air into the hot gas stream. In conventional arrangements, the supply air at the compressor discharge pressure ($P_3$) satisfies the requirements for vane cooling, with the pressure loss in the burner balancing the coolant pressure loss in the vane cooling passages. In a fuel-cooled thermal management system, the impact of the added cooling-air pressure loss across the CHER and external plumbing must be examined with regard to use in the turbine first-stage vanes, which require air supply pressure in excess of the total pressure of the burner discharge.

The required supply pressure for the first-stage vane must match the cycle requirement imposed by the burner pressure loss, while the supply condition at the first-stage turbine blade can be at a substantially lower pressure, because of the flow acceleration and static pressure drop across the vane stage. These requirements imply that air at a single supply pressure does not present a viable design approach, and innovation is required to satisfy the first-stage vane cooling problem. To satisfy the increased pressure requirement, an approach involving additional compression of the vane cooling air was evaluated, with the objective of identifying the weight and power trades. A thermal management system architecture, including an auxiliary compressor for pumping vane coolant, was defined, and a simulation was configured to evaluate the system. The selected system is shown in Figure 1, illustrating the various components.

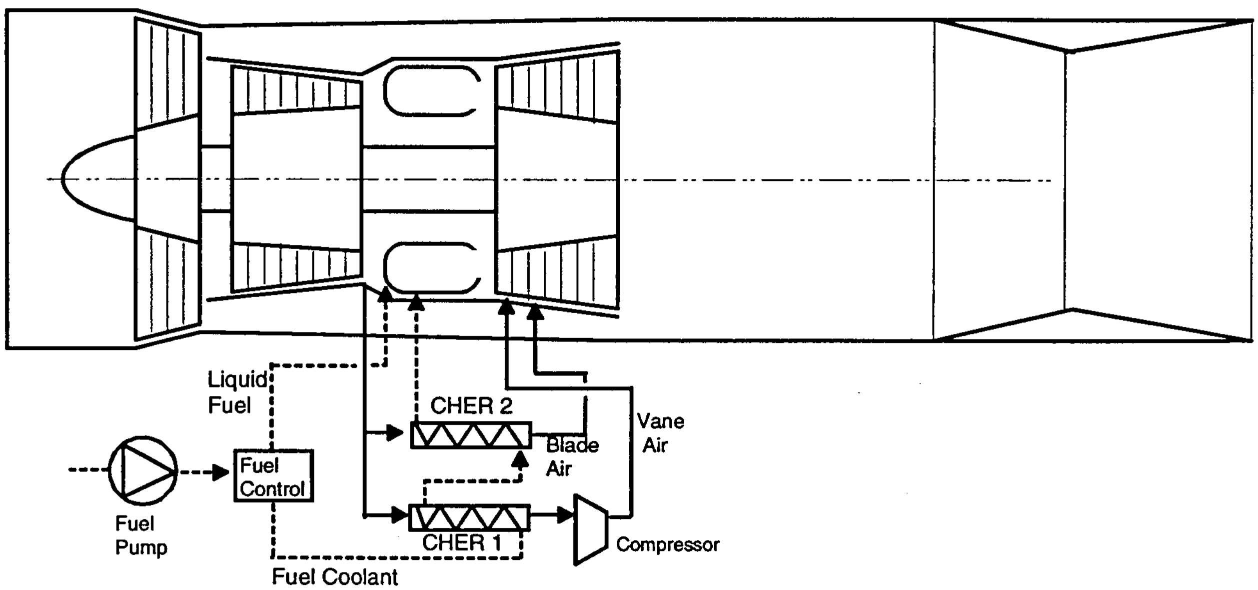


**Figure 1. Concept for Fuel Thermal Management of Turbine Cooling Air**

In this concept, cooling air for the turbine is extracted from the core and cooled in two separate paths. Air for the first-stage turbine vane is cooled and then compressed to a sufficient pressure level for re-insertion into the vane cooling system. The compressor pressure ratio is selected to compensate for pressure loss in the heat exchanger and attendant plumbing system. Power to drive the compressor is extracted from the engine shaft. Cooling air for the blades is not pumped, and is supplied at the pressure level available from the CHER.

Initial evaluation of this arrangement indicated a requirement for a relatively large low-pressure-ratio external compressor, because the vane coolant represented approximately 10% of the core air flow. Therefore, efforts were directed at defining a concept for providing turbine vane coolant with components of minimal size. A more attractive variation of this cooling system was identified, and is illustrated in Figure 2. It entails supercooling a portion of the vane cooling air with fuel, supercharging it using an auxiliary compressor, and mixing it with diffuser bleed air (at $P_3$ and $T_3$) to produce the required amount of coolant at the desired conditions.

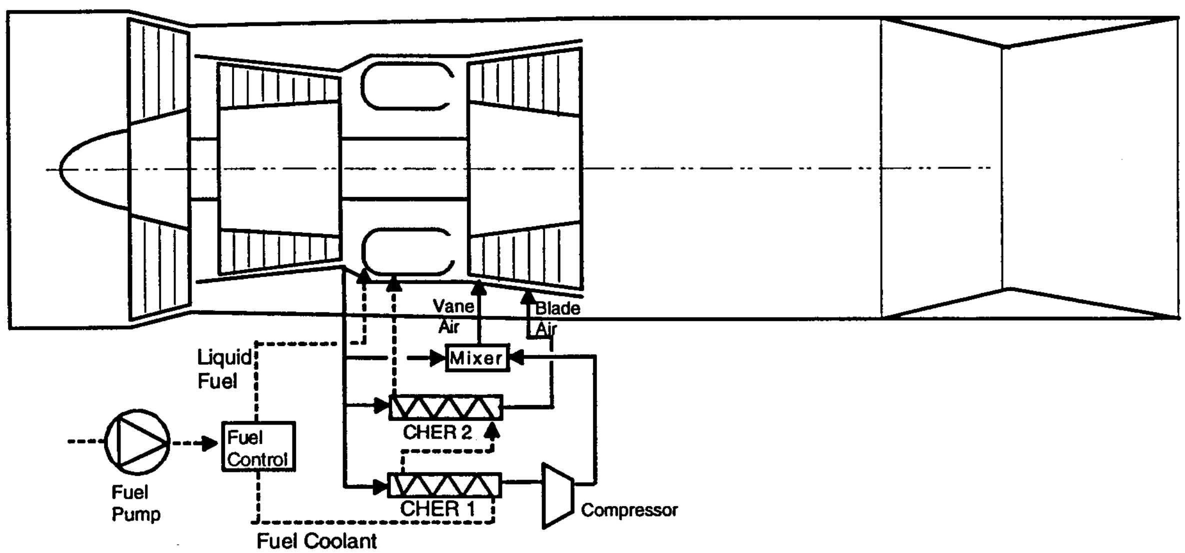


**Figure 2. Improved Concept for Fuel Thermal Management of Turbine Cooling Air**

Preliminary studies indicate that sufficient heat sink is available in the fuel to allow supercooling of a fraction of the airflow, and the net result is a significant reduction in the size and power requirement of the external compressor. Since the cooling system component sizes are directly related to the selected flow split, conceptual design and optimization trade studies of the cooling system were conducted to evaluate the sensitivities of the size, weight, and power demand of the various components.

To design and evaluate the integrated thermal management system illustrated in Figure 2, a model was formulated and a corresponding simulation was developed with the FITSS code. This simulation included the major components in the system, including the heat exchangers, compressor, and mixer required to provide cooled air at the desired conditions. On consideration of the vane coolant mixing requirement, it was decided that a simple mixer would result in unacceptable mixing pressure losses, and that an ejector/mixer would be a more appropriate component. The model of the thermal management system is illustrated in Figure 3, and comprises a series of counterflow heat exchanger modules.

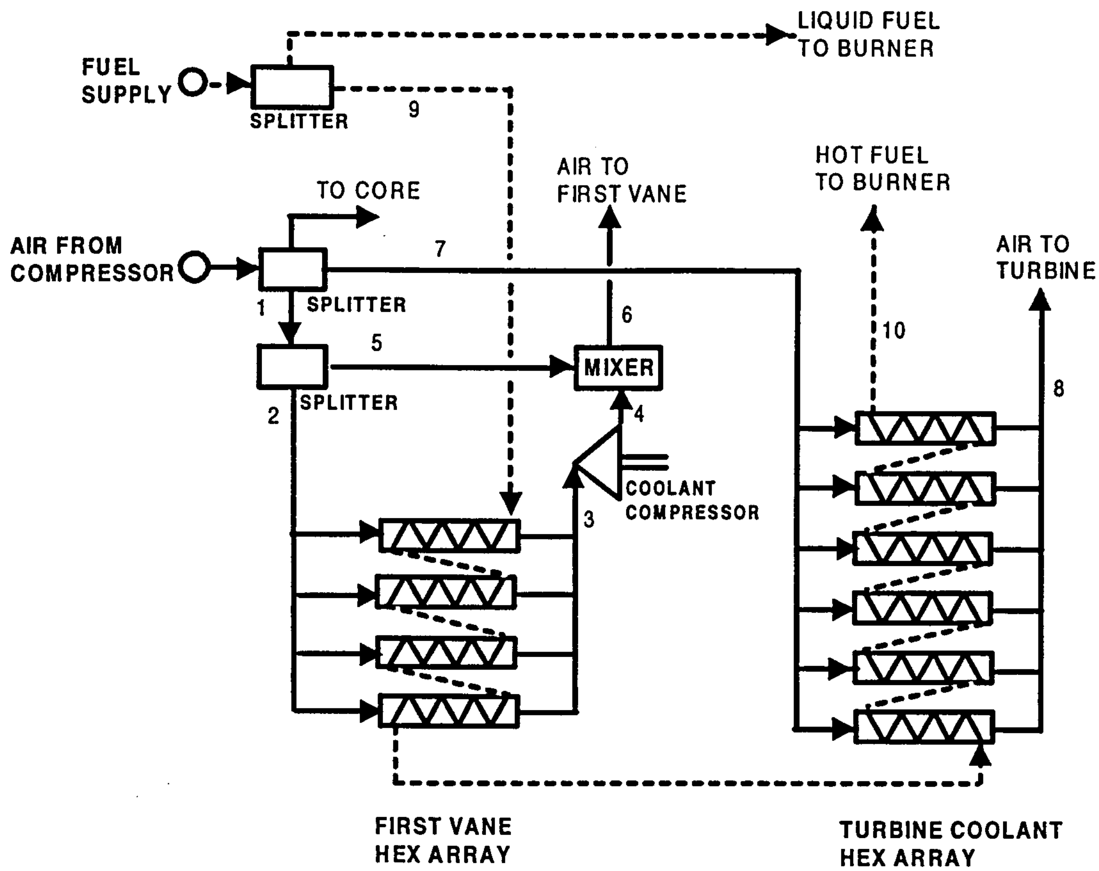


**Figure 3. Conceptual Design Arrangement for Cooling Trade Study**

As shown in Figure 3, a portion of the air for the first turbine vane is cooled and then compressed and mixed with the remainder of the required flow to yield the desired coolant temperature. Splitting the flow and supercooling the pumped fraction allows a reduction of the required pumping power. Another potential system enhancement is to incorporate an ejector into the mixer, to use the cooled stream to pump the hot stream. Considering these various options, it becomes apparent that with several design variables (i.e., flow split, heat exchanger performance and weight, compressor size and power requirements, and performance and weight of the mixer), there is an optimum combination that minimizes system weight/power extraction.

The initial results of the trade study indicated that the required flow split for cooling the first-stage vane is approximately 35% supercooled air mixed with the remaining air at compressor discharge conditions. The pumping requirement was initially estimated to be in the range of 150 - 200 hp, and is dependent on the efficiency of the mixing process. It is assumed that injection and mixing of the cooled and uncooled compressor bleed air streams occurs in a distributed vane ejector, and a model for such an ejector/mixer component was incorporated into the thermal management system simulation.

The thermal management system as illustrated in Figure 3, is composed of a series of counterflow heat exchanger modules arranged in two separate arrays. The first array of four CHER modules provides coolant for the first-stage turbine vanes, while the second array of six modules provides cooled air for the remainder of the turbine. Table 6 summarizes the general details of the heat exchanger designs for achieving a 200°F reduction in coolant temperature in each of the two arrays. The pressure of the coolant supplied to the first-stage vanes is equal to the high-pressure-compressor discharge pressure ($P_3$), while the pressure of the coolant supply to the remainder of the turbine can be up to 10% lower. The local temperatures, pressures and flow rates throughout the thermal management system are tabulated in Table 7 (the locations designated in the table correspond to the numbered locations shown in Figure 3). This set of conditions corresponds to the design-point flight condition of 55,000-ft altitude/Mach number 2.4, with the engine operating at the cruise power level.

The conditions listed in Table 7 include pressure losses due to flow within the heat exchanger passages. However, pressure losses due to flow extraction and re-insertion into the engine are not included and are expected to be an additional 4%. Therefore, summing up all the anticipated losses, the operating requirements of the auxiliary compressor were defined and a conceptual design formulated. It consists of a small single-stage axial-flow fan with ten blades and eleven vanes. The tip radius is 1.55 in., hub-to-tip ratio is 0.322, and the mean airfoil chord is approximately 1.0 in. The shaft speed is 50,000 RPM, with a corrected inlet specific flow of 27.5. At the design point, the power requirement is 107 hp and the efficiency is 84%.

The power supply for the auxiliary compressor has not been identified, but several options seem practical. For example, if this component is integrated with other flight system requirements, the use of a small bleed air turbine appears very attractive. Finally, the total weight of the active heat transfer core elements in the CHERs and the weights of the remaining system components have been estimated (for one engine), based on the use of high-temperature nickel alloys (i.e., Waspalloy, or similar alloy) and diffusion-bonded cores. These approximate weights are shown in Table 8, and include the heat exchanger cores and manifolds, the auxiliary compressor (and the associated power turbine), the air ducting, and the coolant return manifolds for reinserting and distributing cooled air into the engine. Only minimal changes are required in the fuel system and, therefore, the weight increase is negligible as compared to the air system.

**Table 6 - Characteristics of CHER Modules for Turbine Cooling**

| | First-Vane Array | Turbine Array |
|---|---|---|
| Modules | 4 | 6 |
| Material | INCO 625 | INCO 625 |
| Flow length, in. | 11.0 | 4.5 |
| Width, in. | 5.0 | 3.5 |
| Cold Layers | 20 | 25 |
| Hot Layers | 19 | 24 |
| Cold Side (Circular Passages) | | |
| Diameter, in. | 0.040 | 0.040 |
| Pitch, Channels/in | 15 | 15 |
| Wall thickness, in. | 0.020 | 0.020 |
| Hot Side (Fin and Plate) | | |
| Pitch, fins/in | 16 | 16 |
| Fin Height, in. | 0.080 | 0.100 |
| Fin Thickness, in. | 0.008 | 0.008 |
| Weight of HEX cores, lbm | 25 | 10 |

**Table 7 - Thermodynamic Conditions In Cooling System**

| Location | Flow Rate lbm/sec | Temperature °F | Pressure psia |
|---|---|---|---|
| Air for First Vane: | | | |
| 1 | 32.0 | 1200 | 185.2 |
| 2 | 11.2 | 1200 | 185.2 |
| 3 | 11.2 | 571 | 177.1 |
| 4 | 11.2 | 603 | 191.3 |
| 5 | 20.8 | 1200 | 185.2 |
| 6 | 32.0 | 996 | 185.2 |
| Air For Turbine: | | | |
| 7 | 24.7 | 1200 | 185.2 |
| 8 | 24.7 | 991 | 178.4 |
| Fuel: | | | |
| 9 | 7.28 | 200 | 1200 |
| 10 | 7.28 | 785 | 1149 |

**Table 8 - Approximate Weight of Thermal Management System**

| Item | Weight/engine lbs |
|---|---|
| HEX Cores | 160 |
| HEX Manifolds | 26 |
| Turbocompressor | 11 |
| Air Ducts and Manifolds | 163 |
| Total | 360 |

## ECONOMIC ANALYSIS

A first-order economic analysis was performed to estimate the value endothermic technology may bring to the HSCT application. Estimates of weight, specific fuel consumption, manufacturing cost and maintenance cost were used to determine airline direct operating cost plus interest (DOC+I). Airline direct operating cost plus interest (DOC+I) is used to provide a measure of acceptability of new technologies in the airline industry. DOC+I comprises fuel costs, engine maintenance costs, airframe maintenance costs, flight crew costs, hull insurance, depreciation (a function of engine and airframe price) and interest.

Estimates of changes in specific fuel consumption (SFC), engine weight, maintenance cost and purchase price have been made for the new thermal management system concept. Influence coefficients are used to equate each of these changes to a change in DOC+I. DOC+I influence coefficients were generated for a 300 passenger, Mach 2.4 Technology Concept Airplane (TCA) with Mixed Flow Turbofan engines, 3.7 fan pressure ratio and 70% airflow lapse rate (corrected airflow at top of climb/airflow at SL static). A typical 3500 nm economic mission was assumed. Two possible benefits of applying the technology were evaluated, namely to increase lifetime and to improve fuel economy, showing the possible range of costs or benefits this technology may bring to the aircraft application. The influence coefficients are shown in Table 9. They were developed for the HSCT by the Propulsion System Evaluation Team for the HSR program.

The TCA is sized to meet take-off field length, climb and take-off noise requirements for a 5000 nm mission with a subsonic cruise leg which is 15% of the design range. A 3-view drawing of the TCA is shown in Figure 4, and a description of its design mission is shown in Figure 5. Although the HSCT is designed to have the capability to fly 5000 nm in 5.3 hr with a full passenger load, its average mission length on a day-to-day basis will be significantly shorter. A typical 4 hr economic mission of 3500 nm with a 20% subsonic leg was selected in defining the HSCT's airline DOC+I. Economic ground rules for this study are shown in Table 10, and are also traceable to the CPC effort.

**Table 9 - Aircraft Operational Influence Coefficients**

| Parameter | Change | Δ DOC+I |
|---|---|---|
| SFC @ supersonic (M=2.4) | 1% | 0.94% |
| SFC @ subsonic (M=0.9) | 1% | 0.33% |
| weight | 1000 lb | 1.46% |
| maintenance cost | 1% | 0.12% |
| purchase price | $1M | 0.77% |

The following sizing and noise constraints have also been adopted for the Critical Propulsion Components (CPC), Propulsion Systems Integration effort for the TCA to further define the operational design requirements of the HSCT aircraft:

**Sizing Constraints:**

- 10800 ft take-off field length (SL/77°F)
- 300 ft/min minimum rate-of-climb at cruise Mach no.
- 60 min from take-off to top of climb
- 155 knot equivalent air speed at approach
- volume available for fuel (set by wing area)

**Noise Constraints ( FAR Stage III ):**

- -1 db at sideline
- -3 db at cutback
- -1 db at approach

**Table 10 - Ground Rules for Economic Study**

| | |
|---|---|
| Year Dollars | 1995 |
| Fuel Price | $0.63/gallon |
| Utilization | 1030 flights/year |
| Interest rate | 9.00% |
| Economic life | 20 years |
| Resale value | 10% |
| Engine Spares | 23% |

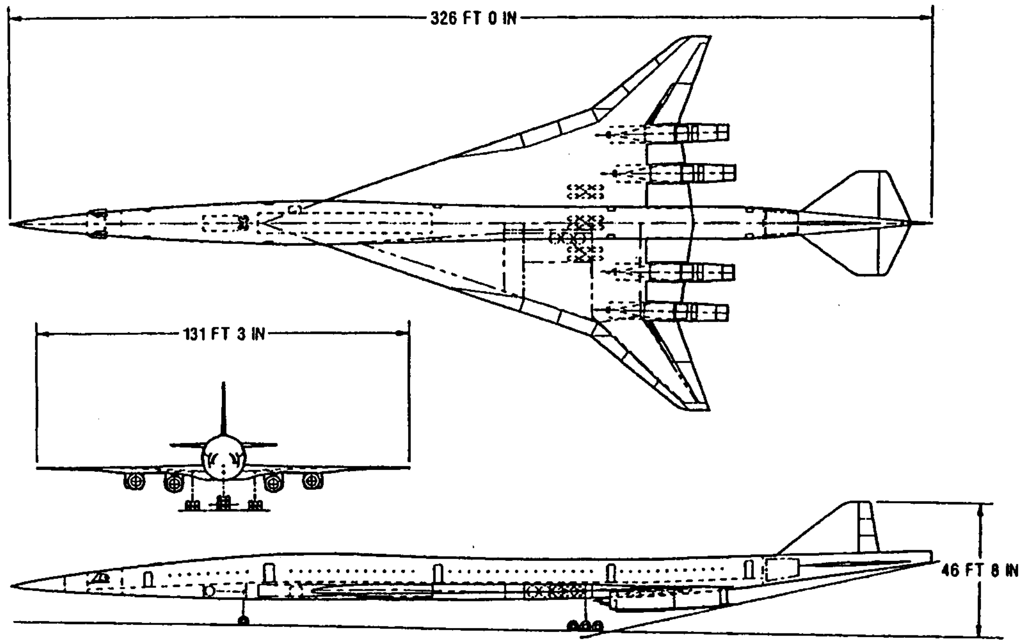


**Figure 4. HSCT Technology Concept Airplane (TCA)**

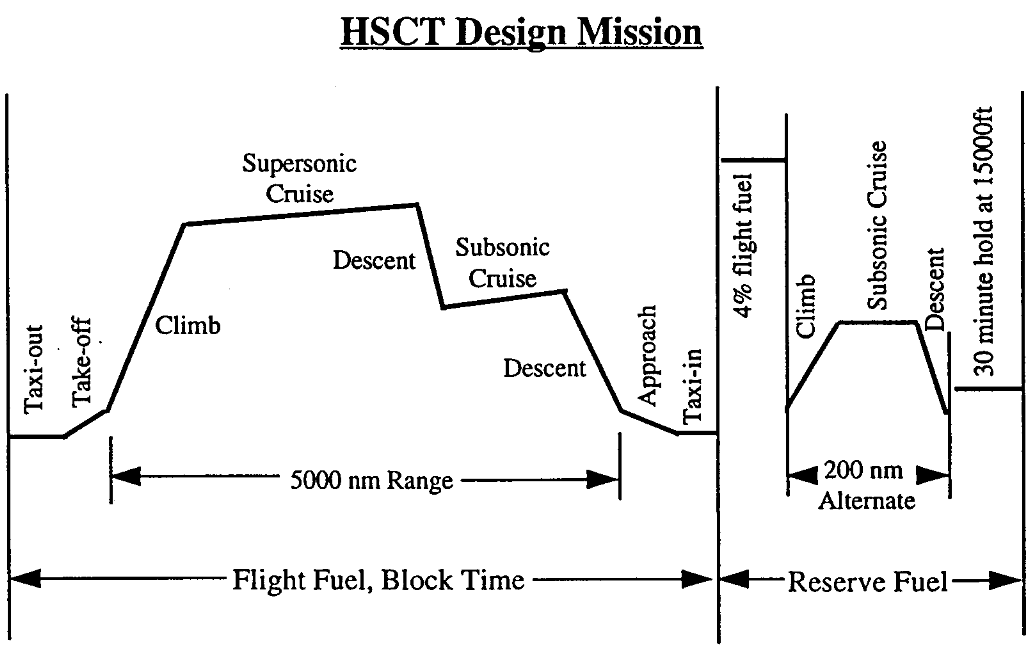


**Figure 5. Technology Concept Airplane Design Mission**

Historical trend data were used to estimate engine cost and maintenance cost. Engine cost estimates are limited to the endothermic cooling system itself and do not include placement and packaging issues which will need to be addressed due to the size of this system. Engine cost impacts are shown in Table 11.

**Table 11 - Estimated Cost of Cooling System Components**

| Component | Weight (lb) | Cost |
|---|---|---|
| heat exchanger cores | 160 | $112000 |
| heat exchanger manifolds | 26 | 91000 |
| turbocompressor | 11 | 110000 |
| air ducts & manifolds | 163 | 65200 |
| | 360 | $197300 |

As stated previously, two methods for applying endothermic reaction technology to the HSCT were considered in this study, namely improved life, and improved fuel economy. The "improved life" approach requires minimal change to the engine cycle. The HSCT has a turbine blade life goal of 9000 hot hours. A 200 F lower metal temperature in the high pressure turbine (HPT) translates into longer blade life and lower engine maintenance cost. Based on maintenance cost trends, a 200 F temperature reduction will result in a 35% increase in the HPT airfoil life, which will effect a 4.5% savings in engine maintenance cost.

The contributors to DOC+I are summarized in Table 12, and rolled up into an overall airline DOC+I for a total system evaluation. The engine weight and purchase price are estimated. The "improved fuel economy" case demonstrates the potential benefits of optimizing the thermal management system with the engine cycle.

**Table 12 - Overall DOC+I for Airline**

| | Improved Life | | Improved Fuel Economy | |
|---|---|---|---|---|
| **Parameter** | **Change** | **Δ DOC+I** | **Change** | **Δ DOC+I** |
| SFC @ supersonic (M=2.4) | -0.1% | -0.09% | -0.7% | -0.66% |
| SFC @ subsonic (M=0.9) | -0.4% | -0.13% | -1.6% | -0.53% |
| Weight | 360 lb | +0.53% | 360 lb | +0.53% |
| Maintenance cost | -4.5% | -0.54% | tbd | tbd |
| Purchase price | $197300 | +0.15% | $197300 | +0.15% |
| **Overall DOC+I** | | -0.08% | | -0.51% |

These data are preliminary, and do not include any maintenance costs associated with the thermal management system (e.g., degradation of the CHER). Because there is presently insufficient information for estimating the thermal management system maintenance requirements, long-duration tests will be recommended as a focus of the research project for validating the concept. However, these initial results suggest that the concept for improving fuel economy has greater potential for economic benefit (due to the HSCT's extreme sensitivity to the TSFC) than does the one for extending the lifetime of the high-pressure turbine. A 0.51% reduction in DOC+I represents a potential saving of $157 million per year/fleet (assuming a 500 airplane fleet), and warrants further study. Although the economic benefit for the "improved lifetime" HSCT application is expected to be small (a 0.08% reduction in DOC+I represents a savings of $25 million per year/fleet), if advanced high-temperature materials are not available for use in the high pressure turbine, then the endothermic cooling system would enable achievement of the life goal. (While the HSCT cycle temperatures are not much different from current generation engines, the time at temperature and stress is up to 30 times greater.) Furthermore, it is important to bear in mind that this analysis was restricted to consideration of fuel cooling approaches for the *current* HSCT engine cycle. The economic benefit would be increased significantly if the cycle could be modified to utilize more of the heat sink available in the fuel. (The approaches considered here take advantage of less than 50% of the fuel heat sink potential.) For example, if critical areas of the compressor were also cooled (in addition to the turbine), then the engine cycle could be operated at higher OPR (pressure ratio) and reduced fuel-air ratio (for $T_4$ = constant).

This first-order economic analysis of the value of endothermic reaction technology in the HSCT indicates that significant monetary savings can be realized due the projected improvement in fuel economy. Although attendant savings would accrue from the corresponding improvement in the life of the high-pressure turbine, the key benefit of fuel cooling to the turbine would be to provide enabling technology for achieving the life goal with available materials.

## RECOMMENDATIONS FOR CONCEPT DEVELOPMENT

The studies conducted in this program have identified several benefits that may be realized by capitalizing on the heat sink capability of current aircraft fuels. Integration of the fuel as a basic resource in the aircraft and propulsion thermal management system has been shown to offer advantages, such as reduced fuel consumption and increased engine life. Achievement of these benefits requires innovation in the approach to thermal management of the engines and airframe. Several critical issues need to be addressed to provide technology readiness of the fuel cooling concepts for application in commercial aircraft. These issues have been considered, and the following critical requirements are identified for additional study:

1. Design of high temperature, high pressure fuel-air heat exchangers with thermal-structural capability adequate to satisfy the safety and long-life requirements of commercial flight systems.
2. Definition of reliable, affordable and maintainable supercritical fuel systems that satisfy the operability requirements of the HSCT application.
3. Expanded use of fuel cooling for improving aircraft performance by increased utilization of the heat sink available in the fuel.

Safety issues are of primary concern, with design goals for components dictating that potential failure modes be fail-safe and fail-operational. Component life requirements for the HSCT will require extended service at near maximum stress conditions. For this reason, structural evaluation of the durability and life expectancy of fuel system components (heat exchangers, control valves, etc.) are recommended as part of the concept development.

Reliability and maintainability issues pertain to the possible degradation of supercritical fuel systems due to formation of coke deposits, and the capability and requirements for maintenance and refurbishment. Evaluation of these issues will require experimental evaluations to define coking sensitivity and deposit rates at the expected operating conditions.

Greater utilization of the cooling capability of the fuel – for example, in the compressor or by integration with the aircraft environmental control system – would result in further benefits to performance and lifetime. Examination of such enhancements should consider individual components as well as the entire engine and the vehicle.

These critical issues are also relevant in military aircraft applications, and some are currently being considered in planning for the on-going and future Integrated High Performance Turbine Engine Technology (IHPTET) demonstration programs. However, these efforts are specific to military applications, and additional studies are needed to address requirements for commercial applications. An initial approach for dealing with these issues and validating the thermal management system concept has been formulated and is described below in a series of tasks that deal with the critical requirements.

### ***Flight-Worthy System Development***

Effective utilization of the fuel as a coolant for air in aircraft systems requires the development of light-weight heat exchangers that are capable of operating safely at high temperatures, pressures and heat fluxes. The process of heating and chemically reforming the fuel must take place rapidly at supercritical pressures to avoid boiling and related flow instabilities and to minimize the formation of coke deposits on the heated surfaces. Therefore, the heat exchangers will operate at very high temperatures and pressures, with large thermal gradients, resulting in potentially high stresses. Structural failure could result in intimate mixing of hot fuel and air, leading to catastrophic failure modes. To be acceptable for use in commercial aircraft, the fuel system components must be robust and designed to provide continued safe operation should a subsystem component fail.

To assure that affordable heat exchangers can be developed to satisfy such demanding requirements, it is recommended that a detailed design of a representative thermal management system be conducted and pursued to an adequate level of fidelity to identify technology capable of satisfying commercial flight operational requirements. This design study should incorporate the following steps:

- Define an engine cycle and representative operational requirements.
- Define a set of thermal requirements for the engine and airframe.
- Design a thermal system architecture to satisfy the defined requirements.
- Evaluate the system across the engine operating envelope to identify the most critical design point.
- Design the system heat exchangers to satisfy the most critical condition. Incorporate structure and fabrication methods being evolved for military application.
- Perform structural evaluation of heat exchangers for both steady state and typical engine transients.
- Assess the ability to satisfy both long-life and cyclic operation requirements.

On completion of the design study, fabricate and test a prototype heat exchanger to demonstrate validity of the design concept. Tests should incorporate representative transient and cyclic operation at appropriate conditions, to demonstrate life capability.

***Operability, Reliability and Maintainability Demonstration***

A fuel-cooled thermal management system for the HSCT must exhibit good performance and operability over the range of flight conditions. It should be able to balance the cooling and propulsion requirements, handle transients, and have acceptable cost, weight and durability. Key challenges to successful implementation are the ability to manage the high temperature fuel delivery system, and to operate with acceptably low coke deposition over the service life of the fuel system components.

Technology programs are currently in progress to identify sensitivity parameters that govern fuel deposit formation, and to define design approaches that may reduce the impact of moderate deposit buildup. Although coke deposition may be somewhat mitigated by the use of appropriate component and system designs, some level of deposit formation is expected to occur with long time operation. This will require the definition of acceptable levels of coke deposition, periodic system maintenance, and methods for in situ component refurbishment. Techniques have been identified that offer potential for in situ cleaning and regeneration of catalytic heat exchangers, and such methods may provide an approach for design of reliable and maintainable thermal management systems.

Therefore, it is recommended that long-duration full-cycle system simulation tests also be performed to determine the maintenance requirements and service intervals, and to demonstrate methods for refurbishment in place after coke buildup. In addition, techniques would be considered for identifying scheduled or unscheduled maintenance requirements.

### ***Expanded Use of Fuel Cooling for Integrated Thermal Management***

Because the HSCT operates at or near the material limits of temperature and stress over a large portion of its flight envelope, it is likely that additional use of the cooling resources of the fuel would produce further benefits in cycle efficiency and engine life. For example, If critical areas of the compressor are also cooled, then the engine cycle could be operated at higher pressure ratios, yielding higher thermal efficiency. However, combustion temperatures must be maintained constant (i.e., f/a must be reduced at high OPR) to avoid increasing $NO_x$ emissions. Therefore, additional cycle studies are required to define such potential cycle benefits. Furthermore, significant benefits may be made to the cooling of the entire vehicle by integrating the aircraft and propulsion system thermal management. Particularly at high speeds, with increasing free-stream total temperature, the fuel becomes a very significant resource for thermal and environmental control of the aircraft. Integration of the aircraft environmental control system with an endothermic thermal management system, as considered for the engine, will provide a significant expansion of the ECS capability, without incurring the performance penalty associated with the use of ram air cooling at high speed. Integration of flight system thermal management is expected to provide an overall benefit in the aircraft performance.

Therefore, an extensive examination of the total requirements for thermal and environmental control of the HSCT aircraft is recommended, with the objective of identifying the potential for combining all on-board thermal requirements into an integrated thermal management system, capitalizing on the cooling potential of both the available engine air and fuel streams. Potential options for thermal system architecture would be examined and evaluated, and the most promising concept for a fully integrated system identified. This selected system should be evaluated to a level identifying potential performance benefits and technology requirements necessary to be ready for application in the HSCT aircraft.

## CONCLUSION

This study identifies the benefits that can be achieved if the high temperature endothermic heat sink capability of Jet A fuel is exploited to provide additional cooling for the engines in the HSCT and AST aircraft. Significant improvements in fuel economy are shown for both aircraft, and increased engine life can be achieved in the HSCT by reducing material temperatures. The economic impact of these benefits on the HSCT show net reductions in direct operating costs accrued by the use of fuel cooling, the specific amount depending on the method of applying the cooling potential to the engine system. Moreover, the payoffs could be increased if the engine cycles were modified to take full advantage of all the heat sink that is available in the fuel.

Several critical issues have been identified with regard to application of endothermic reaction technology in the HSCT aircraft. These include (a) development of high temperature, high pressure fuel-air heat exchangers with thermal-structural capability adequate to satisfy the safety and long-life requirements of commercial flight systems, and (b) definition of reliable, affordable and maintainable supercritical fuel systems that will satisfy the operability requirements of the HSCT application.

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1. Glickstein, M. R., and Spadaccini, L. J., Applications of Endothermic Reaction Technology to the High Speed Civil Transport, *Proceedings of the 1997 JANNAF 34$^{th}$ Combustion Subcommittee, Propulsion Systems Hazards Subcommittee, and Airbreathing Propulsion Subcommittee Joint Meeting,* West Palm Beach, Florida, October 1997.

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The success of strategies for controlling emissions and enhancing performance in High Speed Research and Advanced Subsonic Technology applications may be increased by more effective utilization of the heat sink afforded by the fuel in the vehicle thermal management system. This study quantifies the potential benefits associated with the use of supercritical preheating and endothermic cracking of jet fuel prior to combustion to enhance the thermal management capabilities of the propulsion systems in the High Speed Civil Transport (HSCT) and the Advanced Subsonic Transport (AST). A fuel-cooled thermal management system, consisting of plate-fin heat exchangers and a small auxiliary compressor, is defined for the HSCT, integrated with the engine, and an assessment of the effect on engine performance, weight, and operating cost is performed. Critical requirements for system development are identified, and an approach for validating the concept is described.

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